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FLIGHT-SERVICE EVALUATION OF COMPOSITE STRUCTURAL COMPONENTS

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These longtime flight-service programs will help provide the necessary information required by commercial airlines to commit advanced composites to aircraft structures with confidence. Results of these programs will provide information concerning the stability of composite materials when subjected to various flight environments.

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SUMMARY

A review of the Langley Research Center sponsored programs aimed at flight-service evaluation of composite materials in various applications is presented. These flight-service programs are expected to continue for up to 5 years and include selective reinforcement of an airplane center wing box and a helicopter tail cone and composite replacements for commercial aircraft spoilers and fairings.

These longtime flight-service programs will help provide the necessary information required by commercial airlines to commit advanced composites to aircraft structures with confidence. Results of these programs will provide information concerning the stability of composite materials when subjected to various flight environments.

INTRODUCTION

For the past 10 years numerous research programs have been conducted by the United States government and industry to study the potential of advanced composite materials for use in aircraft structures. Through design, fabrication, test, and analysis these studies have shown that the use of high-stiffness, high-strength composites, such as boron/epoxy and graphite/epoxy, can reduce the mass of structural components by up to 50 percent and thus improve structural efficiency.

Although the structural advantages of composite materials have been demonstrated in the laboratory, widespread applications of composites to commercial aircraft have not occurred. The high cost of composites has been a definite deterrent to their use. Certainly the aircraft manufacturers and the airlines must have confidence in composite materials before they can be committed on a large scale to aircraft primary structural components. The only way actually to substantiate the integrity of composite materials is to investigate their behavior under typical service-load and longtime environmental conditions. As the cost of composites decreases, broadened applications will occur.

A review of the Langley Research Center sponsored programs aimed at flight-service evaluation of composite materials in various applications is presented herein.

These flight-service programs, which are listed in table 1, are expected to continue for up to 5 years and to include selective reinforcement of an airplane center wing box and a helicopter tail cone and composite replacements for commercial aircraft spoilers and fairings. The largest program involves the selective reinforcement of the skins and stringers of the Lockheed C-130 center wing box with uniaxial boron/epoxy. One wing box will be fatigue and static tested, and two wing boxes will be installed on two Air Force aircraft for extended flight-service evaluation. Flight service is scheduled to start in August 1974. The second program involves reinforcement of the longitudinal stringers of a Sikorsky CH-54B helicopter tail cone with uniaxial boron/epoxy. This helicopter has been delivered to the U.S. Army and is in fleet operation with other CH-54B helicopters. The third program is the evaluation of 118 graphite/epoxy spoilers on 27 Boeing 737 airplanes flying throughout the world. Flight service is scheduled to begin in July 1973. The last program involves the installation of 18 PRD-49/epoxy external fairings on Lockheed L-1011 aircraft. Flight service was started in January 1973. Each program is discussed in detail in subsequent sections of the paper.

LOCKHEED C-130 CENTER WING BOX

The Lockheed C-130 transport airplanes have experienced a rapid accumulation of fatigue damage due to severe loading conditions combined with high utilization rates in southeast Asia. Cracking has occurred in the wing skin covers around access doors, doubler termination points, and fuel filler openings. The fatigue life of the C-130 center wing box has been lengthened by increasing skin and hat stiffener areas to reduce overall stress levels, and the C-130 fleet is being retrofitted with the strengthened wing boxes.

In a recent study (ref. 1) it was determined that about 227 kg (500 lbm) of uniaxial boron/epoxy bonded to the skins and stringers of the wing box can reduce the stress levels and thus increase the fatigue life as much as the strengthened aluminum retrofit design. In addition, a 13 percent mass savings was predicted. A photograph of a C-130 center wing box is shown in figure 1. The box is 11.2 m (36.7 ft) long and has a chord of 2.0 m (6.7 ft). The strengthened aluminum box has a mass of about 2177 kg (4800 lbm), whereas the proposed boron/epoxy reinforced box has a mass of 1905 kg (4200 lbm).

The preceding study was followed by the study of references 2 and 3 which consists of fabricating three boron/epoxy reinforced wing boxes, one for ground testing and two for installation in C-130 airplanes that will be flown in regular Air Force service. This program consists of the following five phases: (1) Advanced Development, (2) Detail Design, (3) Fabrication, (4) Ground and Flight Acceptance Tests, and (5) Flight Service and Inspection. The Advanced Development phase (ref. 2) wherein several large components were fabricated and static and fatigue tested has recently been completed.

The Detail Design phase has also been completed and results are presented in reference 3. Accomplishments of these two phases of the program are discussed in the following section.

Design Criteria

Design criteria for the boron/epoxy reinforced wing box had to be established, and considerable effort was required to investigate adhesive bonding and joining techniques before large components could be fabricated. The following design philosophy was adopted for this program:

Maintain C-130E ultimate strength and stiffness

Maintain C-130 B/E aircraft fatigue endurance of 40 000 flight hours or four lifetimes. (C-130 B/E is designation for aircraft retrofitted with strengthened aluminum center wing box.)

Maintain C-130E limit-load capability with no boron/epoxy reinforcement

Maintain C-130 B/E configuration in skin cutout areas

Maintain C-130 B/E rib and spar configurations

The selection of the aluminum/boron area ratio for the center wing box was predicated on a workable balance of four criteria: reduction in mass, equivalent ultimate strength, equivalent damage tolerance, and equivalent fatigue endurance of the C-130 B/E wing box. An 80/20 ratio of aluminum to boron/epoxy was selected to meet these criteria.

Residual Thermal Stresses

Residual thermal stresses were found to have considerable effect on the fatigue endurance of the boron/epoxy reinforced structure. Manufacturing methods were therefore developed to reduce residual stresses.

Schematics of the different bonding techniques investigated are shown in figure 2. In the uppermost sketch, the composite and metal are allowed to expand freely during heating to cure temperature in an autoclave. Once the adhesive is fully cured, the composite and metal are locked together, and upon cooling to room temperature residual thermal stresses are induced in the composite and metal. Because of the lower coefficient of expansion of the composite, compressive stresses are induced in the composite and tensile stresses are induced in the metal. Under certain temperature conditions and stiffness ratios of the adherends considerable warpage can occur. The second technique, shown in figure 2, involves a restrained autoclave cure. The sketch shows the metal structure being restrained by stops fastened to a tool which in this case was steel. The steel tool is heated in the autoclave, and the restraining force is relieved somewhat

by the expansion of the steel. Upon heating, the composite is allowed to expand freely whereas a compressive force is induced in the metal component. Upon cooling, the aluminum has a smaller residual stress and less warpage. The last technique shown is called the "cool tool" method (ref. 2). This method provides maximum constraint to the aluminum adherend as it is heated to bonding temperature while allowing the composite to expand freely. Heating to cure temperature is accomplished by the use of heating blankets applied to the aluminum and composite adherends. Pressure is applied to the composite by air bags. The temperature of the steel tool is maintained close to room temperature by the use of an insulating material. Residual thermal stresses are lower than those obtained with the restrained autoclave cure. In fact, residual compressive stresses can be obtained in the metal adherend at room temperature. This technique not only permits a further improvement of fatigue endurance and/or mass savings but also substantially reduces the structural warpage experienced with other bonding methods. The "cool tool" method will be developed further for use in the fabrication phase of this program.

Composite-to-Metal Load Transfer

Another problem area associated with composite-reinforced metal designs is the transfer of load from the composite to the metal component. Figure 3 shows a typical load-transfer region wherein loads are transferred from the outboard wing to the center wing box at wing station 220.00. In this configuration the existing rainbow fittings and splice straps were maintained. The wing skins are machined to the C-130 B/E configuration in the joint area and are reduced in thickness to the thinner C-130E configuration inboard of the joint. The stiffeners are also machined with the C-130 B/E configuration at the 220.00 joint and are tapered to the C-130E configuration. The laminates for both the wing skins and stiffeners are designed to taper out in the transition area of the C-130E skin to the C-130 B/E skin thickness at the rainbow fitting. This results in an all-metallic joint in the rainbow fitting area and a composite reinforced structure at the net section area of the wing skin. Titanium shims are interleaved in the composite laminates at the ends to increase the bearing strength of the joint.

Component Testing

Compression panels, tension panels, and composite-to-metal load-transfer components were fabricated and subjected to static and cyclic loading to determine strength and fatigue life. Photographs of the three components and their location in the wing box are shown in figure 4. The compression buckling panel shown is representative of a single bay between rib supports; consequently, lateral support was not required. The panel is 51 cm (20 in.) wide and 191 cm (75 in.) long. The tension fatigue panel shown is 102 cm (40 in.) wide and 356 cm (140 in.) long and includes the rainbow fitting at wing

station 220.00 and an access door. The composite-to-metal load-transfer components included the rainbow fittings and were 30 cm (12 in.) wide and 102 cm (40 in.) long. Although the completed wing box will be tested to only four lifetimes, the test components were tested to six and eight lifetimes. The test results are summarized in table 2, and the indication is that, in general, the composite reinforced metal components performed as anticipated.

The fabrication phase of the C-130 program is currently underway and will be followed by ground testing in 1973. As stated previously, flight service is planned for August 1974. The major benefit of this program is expected to be the longtime flight-service experience with composite reinforced materials in a major primary aircraft structure. Flight-service experience should increase confidence in composites and broaden their use in commercial aircraft.

SIKORSKY CH-54B TAIL CONE

The structural design of large helicopter airframes involves both strength and stiffness. The stiffness requirements arise from the necessity of tuning the airframe to prevent amplification of the rotor vibratory forces due to resonance. Thus after providing the strength requirements for flight and ground conditions, it is often necessary to add additional material to increase the natural frequency and prevent resonance of the airframe.

The original Sikorsky CH-54B skycrane airframe, shown in figure 5, was found to be in partial resonance with the main rotor cyclic forces under particular combinations of slung cable length and load which resulted in a "vertical bounce." As shown in the lower left portion of figure 5, thick aluminum skins were added on the top and bottom of the tail cone to increase the vertical bending stiffness. This modification added 59 kg (130 lbm) to the tail-cone mass.

A preliminary analysis showed that bonding 14 kg (30 lbm) of uniaxial boron/epoxy strips to aluminum stringers (see fig. 5) would achieve the same stiffness requirements. In addition, minimum modifications would be required since conventional riveting could be used to attach the composite reinforced stringers to the skins. Langley Research Center and the Langley Directorate, U.S. Army Air Mobility R&D Laboratory jointly sponsored this reinforcement concept under contract (refs. 4 and 5). The study was conducted in two phases. The first involved design and analysis of the tail cone, development of fabrication techniques, and static and fatigue component testing. The second phase involved fabrication of the boron/epoxy reinforced tail cone, vibration and flight testing, and field-service evaluation. The composite reinforced helicopter has been in service in the U.S. Army since April of 1972.

Design and Analysis

Because this is a flight-test program and the use of boron/epoxy reinforced stringers is relatively new, a decision was made to retain the design static strength of the CH-54B tail cone assuming no boron/epoxy reinforcement was present. The criteria set forth for the boron/epoxy reinforcement stipulated that the stiffness of the current production aircraft had to be maintained and that the manufacturing changes to the production aircraft had to be minimal. The stiffness requirement was met by bonding 50-ply boron/epoxy strips 1.91 by 0.64 cm (0.75 by 0.25 in.) to the vertical legs of the standard aluminum stringers. Twelve stringers were reinforced with boron/epoxy, seven on the lower skin and five on the upper skin. This reinforcement permitted skin-gage reductions from 0.356 cm to 0.102 cm (0.140 in. to 0.040 in.) in some sections of the tail cone. Figure 6 illustrates the layout of the tail cone and stringer locations. To meet the design static-strength requirements with no boron/epoxy reinforcement, short stringer sections had to be added to reduce skin buckling. These short stringer sections are referred to as panel breakers in figure 6.

Thermal Stresses

The analysis of the composite reinforced structure included the induced fabrication and environmental thermal effects. The thermal effect of fabrication is to induce tension stresses in the aluminum stringers and compression stresses in the boron/epoxy reinforcement at any temperature below the curing temperature of 394 K (250° F). The environmental thermal effects are based on the stress-free temperature of 394 K (250° F) for the reinforced stringers, the room temperature stress-free condition for the aluminum skins, and the operating temperature for the structure. Thermal stresses are maximum at 219 K (-65° F), the minimum environmental temperature for the CH-54B.

An analytical investigation showed that the induced thermal loads were well within the allowable loads for the skin-stringer combinations, and positive margins of safety existed in all flight conditions. Warpage did not seem to be a problem; hence, no attempt was made to develop fabrication techniques to alleviate residual thermal stresses.

Since the margins of safety were based upon calculated induced thermal stresses, it was felt that some experimental verification was necessary. A test program was conducted to determine the actual induced stresses in the boron/epoxy reinforced stringers, to establish design coefficients of thermal expansion for aluminum and boron/epoxy, and to verify the analysis for the predicted thermal stresses.

An aluminum stringer and a boron/epoxy strip were instrumented with strain gages and bonded together. The instrumentation was monitored as the bonded piece was cooled from the bonding temperature of 394 K (250° F) to the minimum service temperature of 219 K (-65° F). An additional "dummy" aluminum stringer and a boron/epoxy strip were

instrumented with strain gages and monitored over the same temperature range. The "dummy" instrumentation was used to determine both the thermal coefficients of expansion for the aluminum and boron/epoxy, and the temperature-induced apparent strain corrections for the strain gages used. The gages bonded to the boron/epoxy reinforced stringer were used to determine the thermal strains induced in the aluminum and boron/epoxy constituents. The resulting thermal strains, corrected for temperature-induced apparent strains, are shown in figure 7 as a function of temperature. In general, the test program verified the analytical strain predictions.

Composite-to-Metal Load Transfer

An important aspect of the design phase of the program was the selection of a joint for suitable load transfer between the aluminum stringer and the boron/epoxy reinforcement. Since this application of boron/epoxy is for a stiffness design, the boron/epoxy strips can be stepped down to an all-metal joint as the metal is capable of full strength requirements. The design was critical in that it had to be such that the induced shear-stress peaks in the adhesive bond were acceptable. The boron/epoxy strips must be continuously bonded to the aluminum to stiffen the tail cone effectively. A design concept of incorporating a fiberglass insert at the beginning of a tapered joint was investigated. The insert had the effect of introducing a "soft" joint end, thereby reducing the sudden discontinuity in stiffness. A Sikorsky developed composite-joint program showed that the peak-induced shear stresses were reduced by about 50 percent below those in a joint without a fiberglass insert. The adhesive shear-stress distribution for a 31.8-cm (12.5-in.) tapered joint subjected to actual flight loads is shown in figure 8. Two layers of 0° fiberglass/epoxy (1002-S) were introduced over the first 5.1 cm (2.0 in.) of the joint between the boron and aluminum as illustrated in the top of figure 8. The computed shear-stress distribution from an approximate analysis indicates a series of rapid stress-gradient changes due to discontinuities produced by the introduction of each additional layer of composite. The continuous curve is faired to represent what might be expected from practical considerations. The maximum shear stress for the dashed curve is about 7600 kPa (1100 psi). This particular joint geometry was selected for the boron/epoxy reinforced stringers installed in the flight-test article. The above stress level is within the allowable shear strength of the adhesive used in this program.

Tension, Compression, and Shear Tests

As stated previously, the stiffness of the current-production aircraft had to be met with the composite reinforced structural configuration. Tensile specimens 15 cm (6 in.) wide and 61 cm (24 in.) long with one boron/epoxy reinforced stringer attached were fabricated for static and fatigue testing. The load-strain curves for the reinforced and nonreinforced stringers are shown in figure 9. The strains shown are the strains at the

neutral axis of the reinforced section. The following conclusions can be drawn from these tests: (1) the tensile strength of the boron/epoxy reinforced stringers is equivalent to that of the nonreinforced stringers, (2) the limiting factor in the load-carrying ability of the reinforced stringer is the bond strength, (3) the experimentally determined area-modulus product for both the reinforced and nonreinforced stringers agreed with analytical results, and (4) the area-modulus product of the reinforced stringer is equivalent to that of the current-production aluminum stringer.

Tension-fatigue tests were conducted to evaluate the effect of repeated loading on both the all-aluminum stringers and the boron/epoxy reinforced stringer structure and to assure that the fatigue life is sufficient for the lifetime of the aircraft. Previous experience shows that the CH-54B aircraft flies 600 to 700 hours per year. The airframe life was thus estimated at 10 000 hours or approximately 15 years of operation. The test loads used were based on measured-in-flight loads and included maximum measured vibratory loads and the maximum ground-air-ground cycle loads. Test results showed that both the all-aluminum and boron/epoxy reinforced panels had a fatigue life that exceeded four times the aircraft-fatigue schedule. It was also determined that the tapered load-transfer region of the boron/epoxy composite appears to be the critical section in fatigue.

Compression and shear tests were conducted to obtain strength data. The shear specimens were reinforced by doublers at the load introductory points, and the compression panels were potted to facilitate load introduction. The boron/epoxy reinforcements were terminated before the potted end to simulate the proposed aircraft construction. Figure 10 illustrates that the boron/epoxy reinforced compression panel failed by local crippling at the end of the tapered reinforcement. The failure load was approximately 40 percent higher than the failure load for the conventional all-aluminum panels. Figure 11 illustrates the shear-panel test apparatus and definitive shear buckles. The shear panels failed at about 90 percent of the predicted load with failure occurring in the riveted-edge attachments.

Flight Tests

The purpose of the flight test was to demonstrate the absence of "vertical bounce" and to measure stress levels at discrete points in the boron/epoxy reinforced tail cone. The flight tests were conducted throughout the normal airspeed and rotor-speed envelope and included level flight and normal maneuvering flight conditions. In attempting to induce "vertical bounce" tests were conducted with a single point load of 11 340 kg (25 000 lbm) at various cable lengths and with one blade of the main rotor positioned 5 cm (2 in.) out of track to provide a rotor-exciting force. "Vertical bounce" could not be induced in the aircraft, and the strain-gage readings indicate that the stresses on the individual structural elements of the boron/epoxy reinforced stringer and the stresses on

the conventional structure are such that the criterion of a 10 000-hour life with a factor of four was exceeded. The test program was successfully completed in February 1972.

Periodic inspection of the tail cone will be made to check for composite disbonds or any other difficulties that may occur. Test methods using portable ultrasonic equipment will be developed and used for these inspections. Figure 12 shows an inside view of the boron/epoxy reinforced tail cone and illustrates the accessibility of the boron reinforced stringers for inspections. No damage has been detected after more than 150 hours of flight service.

The results of this program to date indicate that the use of boron/epoxy reinforced structures is feasible, the boron/epoxy reinforced tail cone is 14 percent lighter than the strengthened aluminum tail cone, and continued development should lead to an early commitment of the selective reinforcement concept to production aircraft.

BOEING 737 SPOILERS

A program to manufacture and place composite spoilers in commercial airline service to determine the longtime behavior under a variety of flight-service-load and environmental conditions is under way. The culmination of this work will be the testing of selected spoilers after flight exposure and correlation of results with tests of materials specimens after ground-based exposures simulating the flight environment. The Boeing 737 airplane was selected for this program because of its worldwide availability, its high use rate of about 3000 flight hours per year, and its spectrum of short flights which will frequently load the spoilers. Figure 13 illustrates the location of the Boeing 737 spoilers. Though important functionally, the spoilers are not critical to the safety of the aircraft. The installation positions for the spoilers proposed for this program are outside the lightning-strike zone so that lightning is not a critical consideration.

The Boeing Company conducted a preliminary investigation in which the flight spoilers of the 737 airplane were selected as components to advance the use of graphite/epoxy materials for structural applications. In this program spoilers were developed with the aluminum skins replaced by graphite/epoxy and the aluminum end ribs replaced by fiberglass. The aluminum hinge fittings, spar, and honeycomb core common to the production spoiler were retained. Figure 14 shows the construction of a 737 spoiler with graphite/epoxy skins. It is noted that the graphite/epoxy spoiler is 15 percent lighter than the all-aluminum production spoiler.

Langley Research Center chose to follow up this program with a comprehensive ground and flight-test program. This two-phase program is being conducted under NASA contract. The first phase involves screening various graphite/epoxy material systems, fabrication and installation of 108 graphite/epoxy spoilers on 27 aircraft for flight service

throughout the world, fabricating three spoilers (one from each material system selected) for ground tests, and fabricating three spares in case foreign-object damage or malfunction of any spoiler occurs. In addition, 900 test coupons will be delivered to Langley for use in longtime ground-based environmental testing. Phase II is concerned with the development of advanced spoilers such that maximum utilization of composites is achieved. Plans call for ten advanced-design spoilers to be flight tested with one advanced spoiler fabricated for ground testing.

Phase I – Graphite/Epoxy Spoilers

Eight different graphite/epoxy material systems were screened to arrive at the three materials to be used for the phase I spoilers. The three material systems selected were as follows: (1) Hercules type AS fiber with Hercules X3501 resin, (2) Thornel 300 fiber with Union Carbide 2544 resin and, (3) Thornel 300 fiber with Narmco 5209 resin. The design of the 737 spoiler is stiffness-critical, and the load-deflection characteristics of the graphite/epoxy spoilers must match the aluminum production spoilers. Figure 15 shows load-deflection curves that were generated by Boeing in their preliminary study. Both the Courtaulds HMS and Hitco HMG-50 graphite/epoxy spoilers are slightly stiffer than the 737 production aluminum spoiler. As stated previously, three graphite/epoxy spoilers (one for each material system) will be static tested to failure. Subsequently, several of the 108 flight spoilers will be randomly selected and removed from the aircraft for static testing to compare the structural response with baseline tests. The ground-based environmental tests to be conducted at Langley Research Center will include compression, flexure, and shear specimens. The effects of humidity, thermal cycling, and various other environments will be investigated.

Phase II – Advanced-Design Spoilers

The objective of phase II of the program is to arrive at more advanced versions of the basic spoiler with maximum effective utilization of composite materials, such as chopped-fiber molded parts to replace metal parts, other types of advanced filaments to replace the graphite used in phase I, and a core such as Nomex or PRD-49 to replace the aluminum honeycomb core. An example of a potential advanced molded hinge fitting is shown in figure 16.

As stated previously, ten advanced composite spoilers will be placed in flight service and one spoiler will be static tested. In addition, 300 compression, flexure, and shear coupons will be fabricated and delivered to Langley for environmental exposures.

Primary emphasis of this program is not on mass savings but on advancing the composites technology and building confidence in composite materials through longtime flight service and ground-based environmental testing. Five airlines throughout the world have

tentatively agreed to fly the composite spoilers in a variety of environments and route structures. Also, manufacturing over 100 spoilers should provide a valuable base for determining the cost depreciation or the learning-curve slope of advanced composite parts when fabricated in production quantities.

LOCKHEED L-1011 EXTERNAL FAIRINGS

The Lockheed-California Company, under NASA contract (ref. 6) installed 18 PRD-49/epoxy external fairings on three production wide-bodied L-1011 commercial transports. A ground-based environmental test program will be conducted at Langley Research Center in parallel with the flight-service program. The primary objective of this program is to obtain longtime flight-service experience with the relatively new PRD-49/epoxy composite material in the commercial-airlines environment.

In order to facilitate an early application of PRD-49/epoxy composites, this program was essentially a direct materials substitution for the currently used glass/epoxy fairings. There are three distinctly different fairings installed on aircraft for flight service. The three different locations of the fairings are shown in figure 17.

Wing-to-Fuselage Fairing

The largest panel is a 152-cm by 170-cm (60-in. by 67-in.) wing-to-fuselage fairing. A view of this fairing is shown in figure 18. This fairing (approximately 3 m^2 (30 ft^2) in area) has a slight single curvature and represents one of the largest fairing panels on the L-1011 aircraft. The wing-to-fuselage fairing is subject to aerodynamic loads only and is designed as a "floating" structure with no load transmitted into the fairing from adjacent structures. The design ultimate loads are 8.3 kPa (1.2 psi) ultimate internal pressure and 16.5 kPa (2.4 psi) ultimate external pressure. The outer skin of the PRD-49/epoxy wing-to-fuselage fairing is 0.051 cm (0.020 in.) thick (two plies of 120 fabric and one ply of 181 fabric), and the inner skin is 0.038 cm (0.015 in.) thick (three plies of 120 fabric). The honeycomb core is 48.1 kg/m^3 (3.0 lbm/ft^3) Nomex with 0.3-cm (1/8-in.) cell size and has a thickness of 2.24 cm (0.88 in.). The resin system used is Hexcel F-155 which cures at 394 K (250° F). Each panel has an aluminum flame spray coat on the external surface to discharge electric potential from the fairing surface to the adjacent structure. The panels were fabricated by the Heath-Tecna Corporation, Kent, Washington.

Pressure tests have been conducted on a 152-cm by 170-cm (60-in. by 67-in.) panel. The panel failed at a pressure of 20.34 kPa (2.95 psi) or 125 percent of design ultimate. Maximum panel deflection was 5.1 cm (2.0 in.).

Wing-to-Fuselage Fillet

The wing-to-fuselage fillet panel shown in figure 17 is of solid PRD-49/epoxy laminate construction 0.23 cm (0.090 in.) thick, tapering to 0.076-cm (0.030-in.) thick-edge closeouts. The part as shown in figure 19 is approximately 0.16 m^2 (1.75 ft^2) in area. This panel has the same design loads as given for the wing-to-fuselage fairings. Because of its smaller size, the fillet panel is subject to much less actual deflection and, therefore, the static-load test on the fairing panel will serve to verify this part as well. These panels were also fabricated with the Hexcel F-155 resin system.

Center-Engine Fairing

The center-engine fairing panel shown in figure 17 is a sandwich construction, 0.64 cm thick (0.25 in.), with 0.051-cm-thick (0.020-in.) skins (two plies of 120 fabric and one ply of 181 fabric), and has 0.25-cm to 0.318-cm-thick (0.10-in. to 0.125-in.) solid-laminate edge members. The part shown in figure 20 has an area of approximately 0.7 m^2 (7.5 ft^2). This part, because of the proximity to the center engine, requires a 422 K (300° F) service epoxy resin matrix. The resin selected was Hexcel F-161 which cures at 450 K (350° F).

This fairing was designed to withstand a pressure differential of 6.9 kPa (1.0 psi), which is less than the wing-to-fuselage fairing and fillet panels. As with the other parts, only air loads are encountered, since there are no loads transmitted from adjacent structures. Hence, the static-load test performed on the wing-to-fuselage fairing panel also provides structural verification for the center-engine fairing panels which are smaller and have lighter loads.

Six fairings (one ship set) have been installed on three different aircraft for exposures to different environments and route structures. Mass savings range from 25 to 32 percent, even using the baseline honeycomb-core thickness. Three different airlines have tentatively agreed to fly the PRD-49/epoxy fairings. The panels will be flown for at least 5 years in commercial airline service. Concurrently, 600 test coupons (compression, flexure, and shear) will be exposed to different environments and tested at the Langley Research Center.

This program is a step to establish confidence in the use of PRD-49/epoxy composites through extended commercial-airline service evaluation. Since PRD-49 has a density that is 40 percent less than that of fiberglass and a modulus $1\frac{1}{2}$ times that of fiberglass, an early application of this material could save considerable mass in the next generation of aircraft. Of course, a broadened application is required to make the PRD-49 material cost effective.

CONCLUDING REMARKS

A review of the Langley Research Center sponsored programs aimed at flight-service evaluation of various composite materials has been presented. Design concepts and tests to substantiate the component behavior have been described. These flight-service programs of up to 5 years duration will help provide the necessary confidence required by commercial airlines to commit advanced composites to aircraft structures. Results of these programs will provide information concerning the stability of composite materials when subjected to various flight environments.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., May 21, 1973.

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5. Welge, R. T.: Application of Boron/Epoxy Reinforced Aluminum Stringers and Boron/Epoxy Skid Gear for the CH54B Helicopter Tail Cone. Phase II: Fabrication, Inspection and Flight Test. Contract No. NAS 1-10459, Sikorsky Aircraft, United Aircraft Corp., July 1972. (Available as NASA CR-112101.)

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TABLE I.- LANGLEY RESEARCH CENTER SPONSORED FLIGHT-SERVICE PROGRAMS

Program description	Number of components		Composite material	Number of aircraft involved	Estimated start of flight service
	Ground	Flight			
C-130 reinforced center wing box (refs. 1, 2, and 3)	1	2	Boron/epoxy	2	August 1974
CH-54B reinforced tail cone (refs. 4 and 5)	1	1	Boron/epoxy	1	April 1972
737 composite spoilers NAS1-11668 Boeing	7	118	Graphite/epoxy and others	27	July 1973
L-1011 external fairings (ref. 6)	1	18	PRD-49/epoxy	3	January 1973

TABLE II.- C-130 WING-BOX REINFORCED-COMPONENTS TESTS

Component	Test results
Compression panel	$P_{test} = 0.96P_{design\ ultimate}$ (end failure rather than column buckling)
Tension panel (1)	Fatigue life > 6 lifetimes $P_{residual\ strength} = 1.09P_{design\ ultimate}$
Tension panel (2)	Fatigue life > 8 lifetimes $P_{residual\ strength} = 0.92P_{design\ ultimate}$
Composite-to-metal load-transfer joint	Fatigue life > 8 lifetimes $P_{residual\ strength} = 1.35P_{design\ ultimate}$
Design criteria: Composite reinforced-aluminum components to meet or exceed static strength, fatigue resistance, and damage containment of comparable aluminum components.	

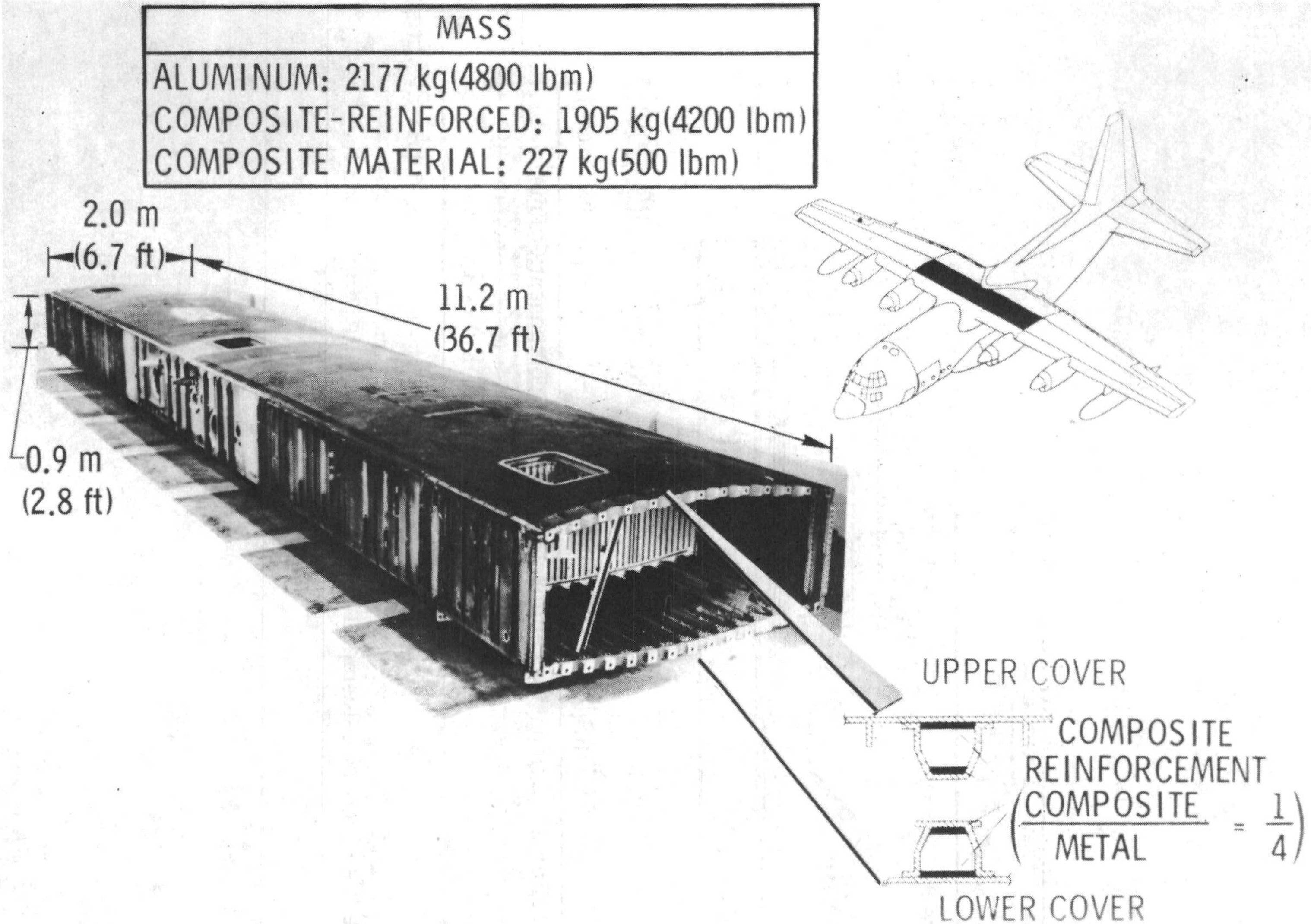

Figure 1.- C-130 center wing box.

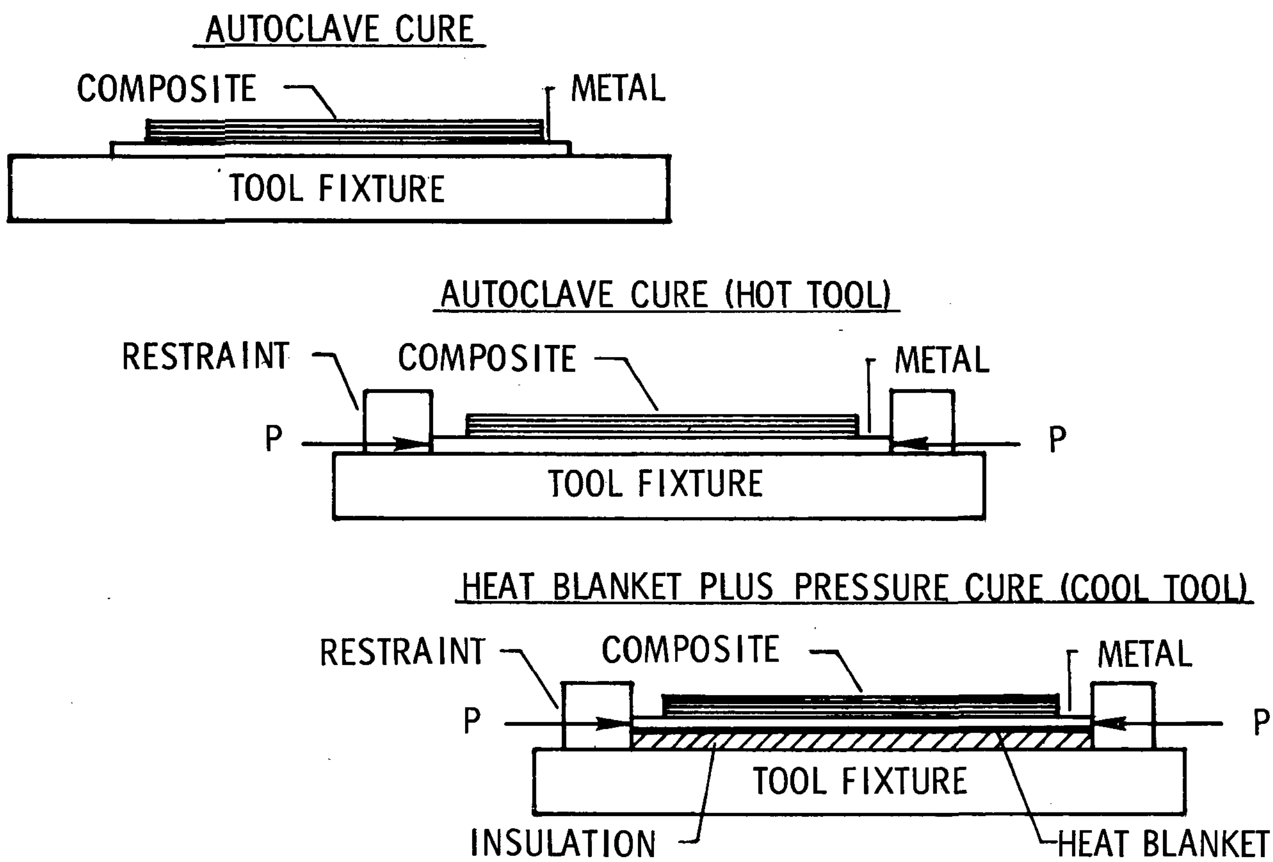

Figure 2.- Composite-to-metal bonding techniques. (P is the load induced during bonding.)

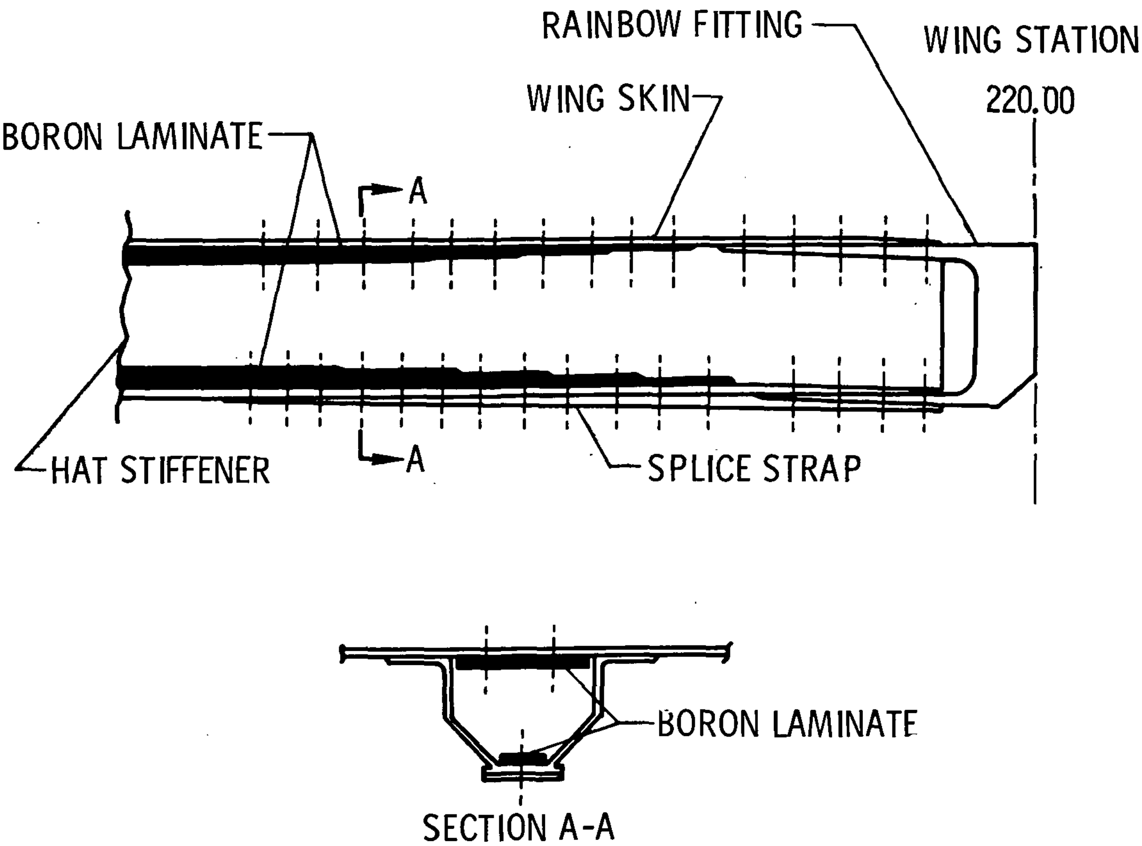

Figure 3.- C-130 wing-box joint design.

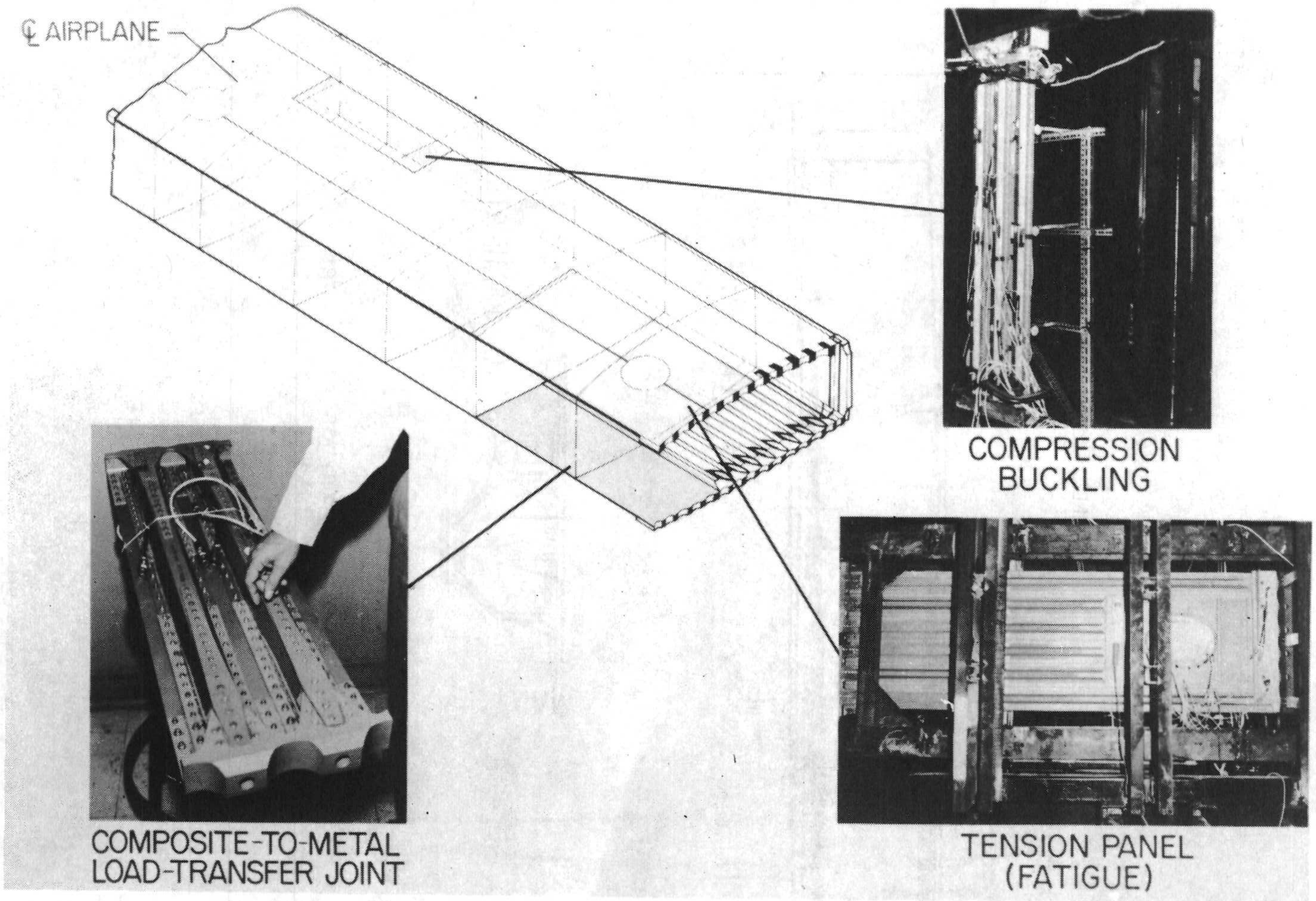

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Figure 4.- C-130 composite reinforced wing-box test components.

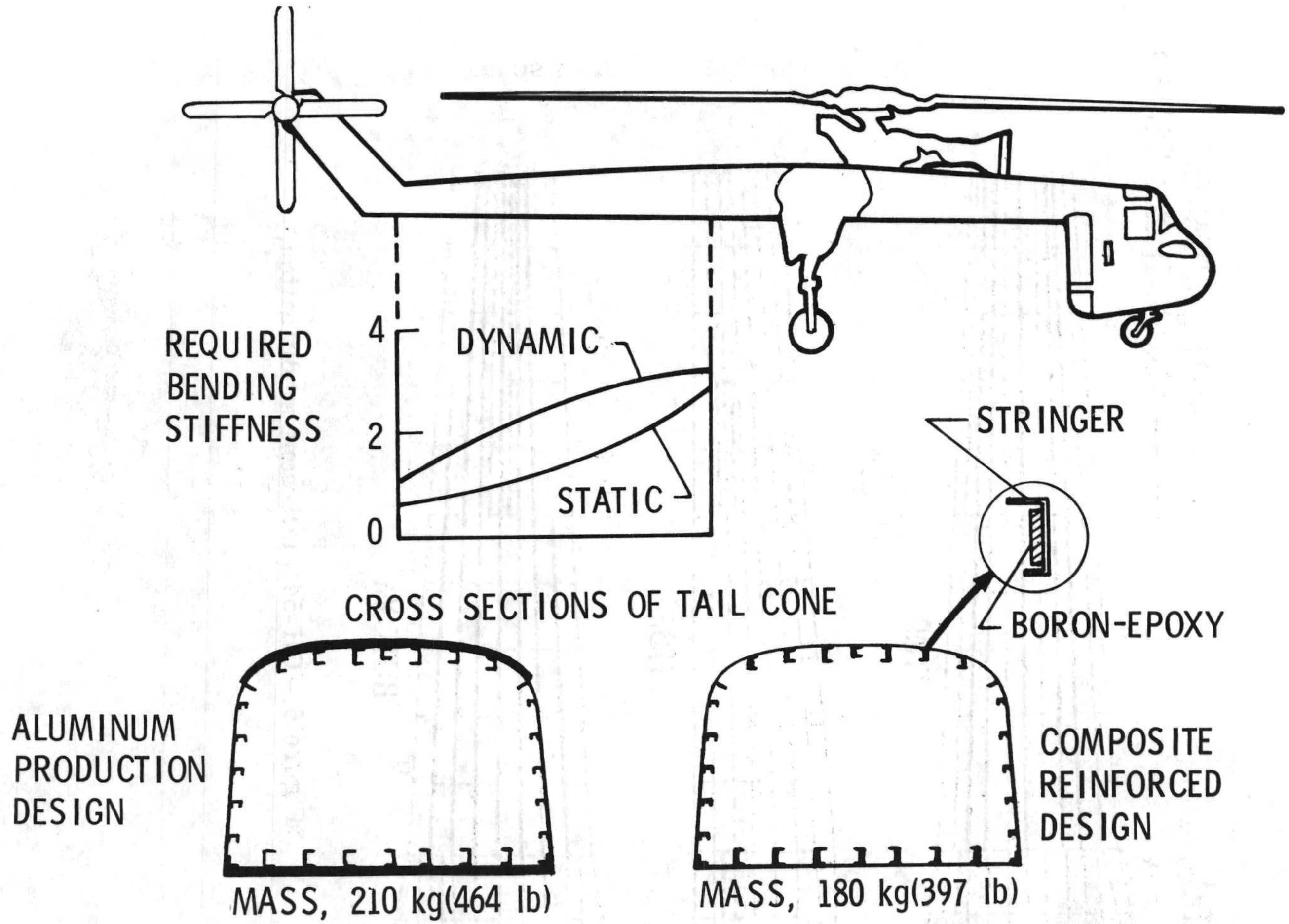


Figure 5.- Sikorsky CH-54B helicopter reinforced tail cone.

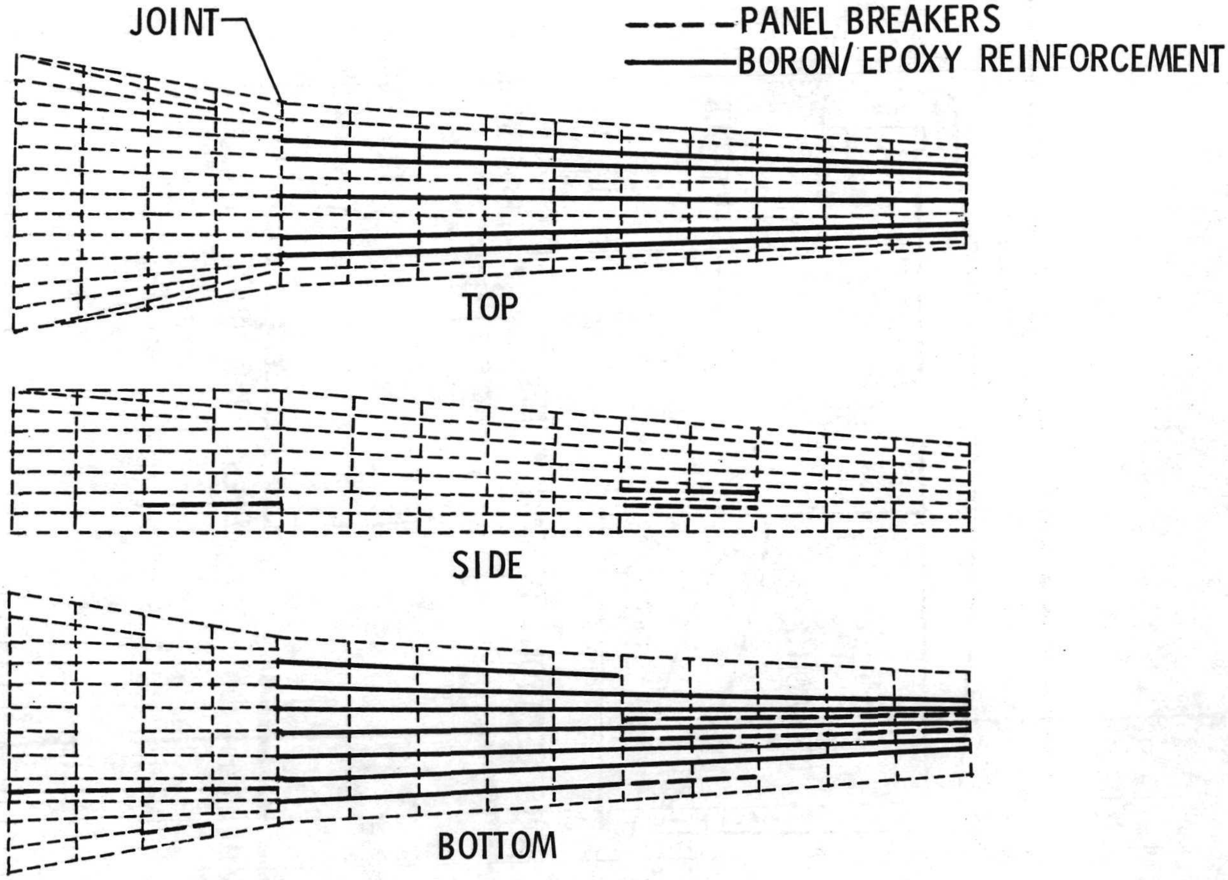


Figure 6.- CH-54B tail-cone modification.

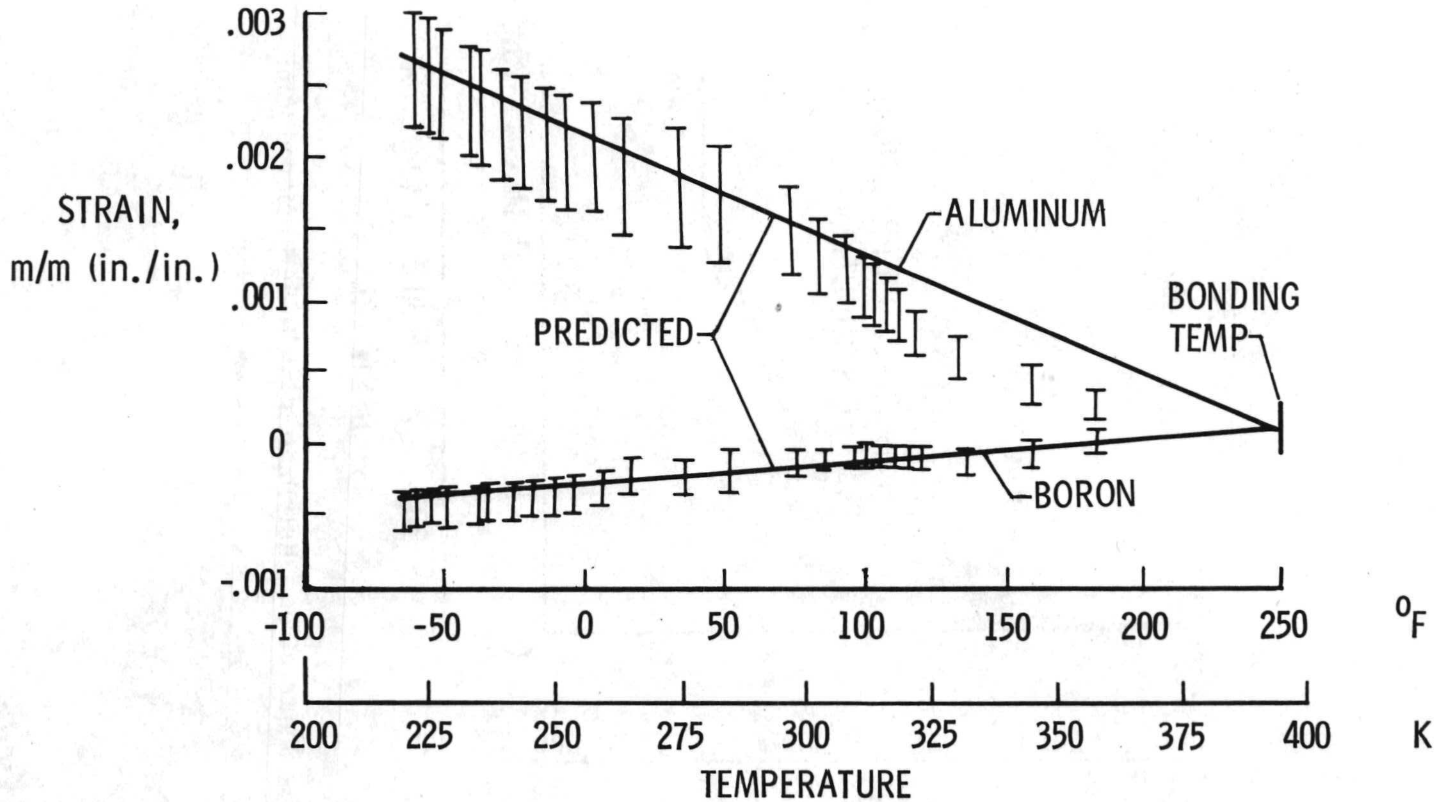


Figure 7.- Induced thermal strains in CH-54B reinforced stringers.

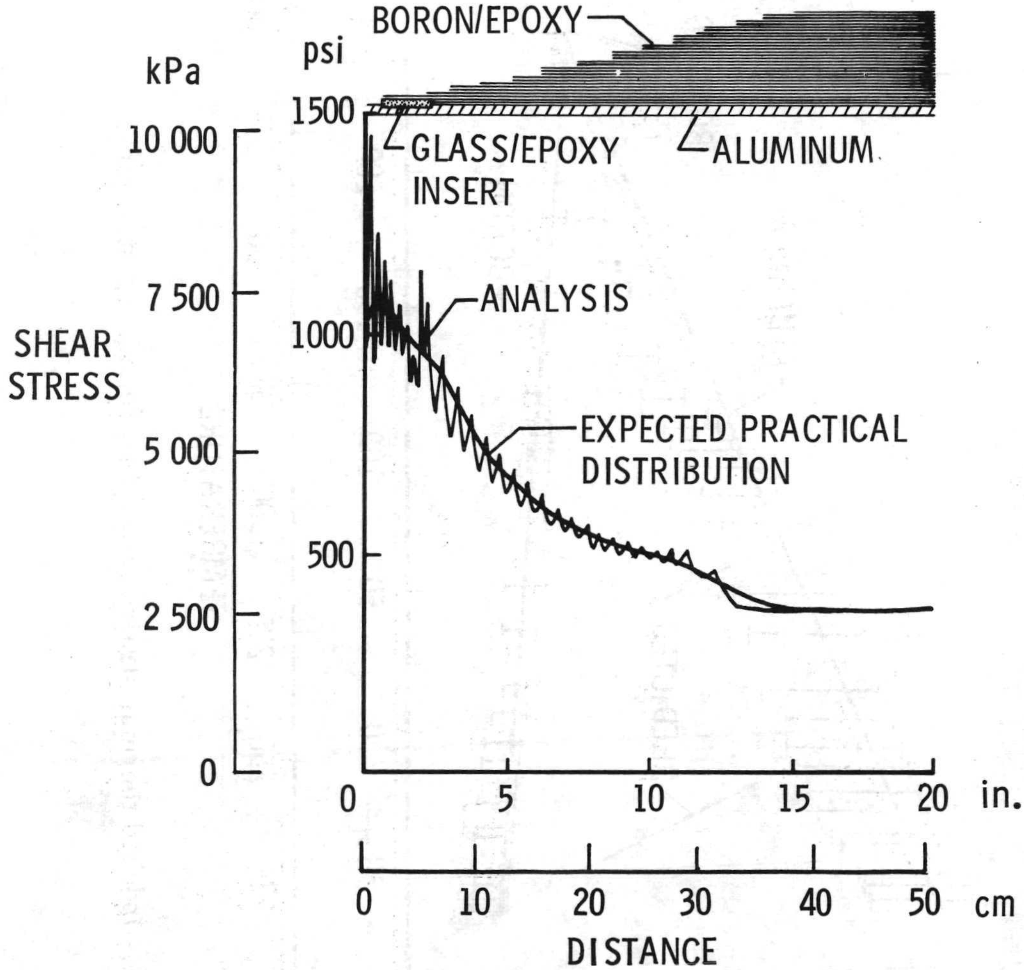


Figure 8.- Shear-stress distribution in CH-54B tapered joint.

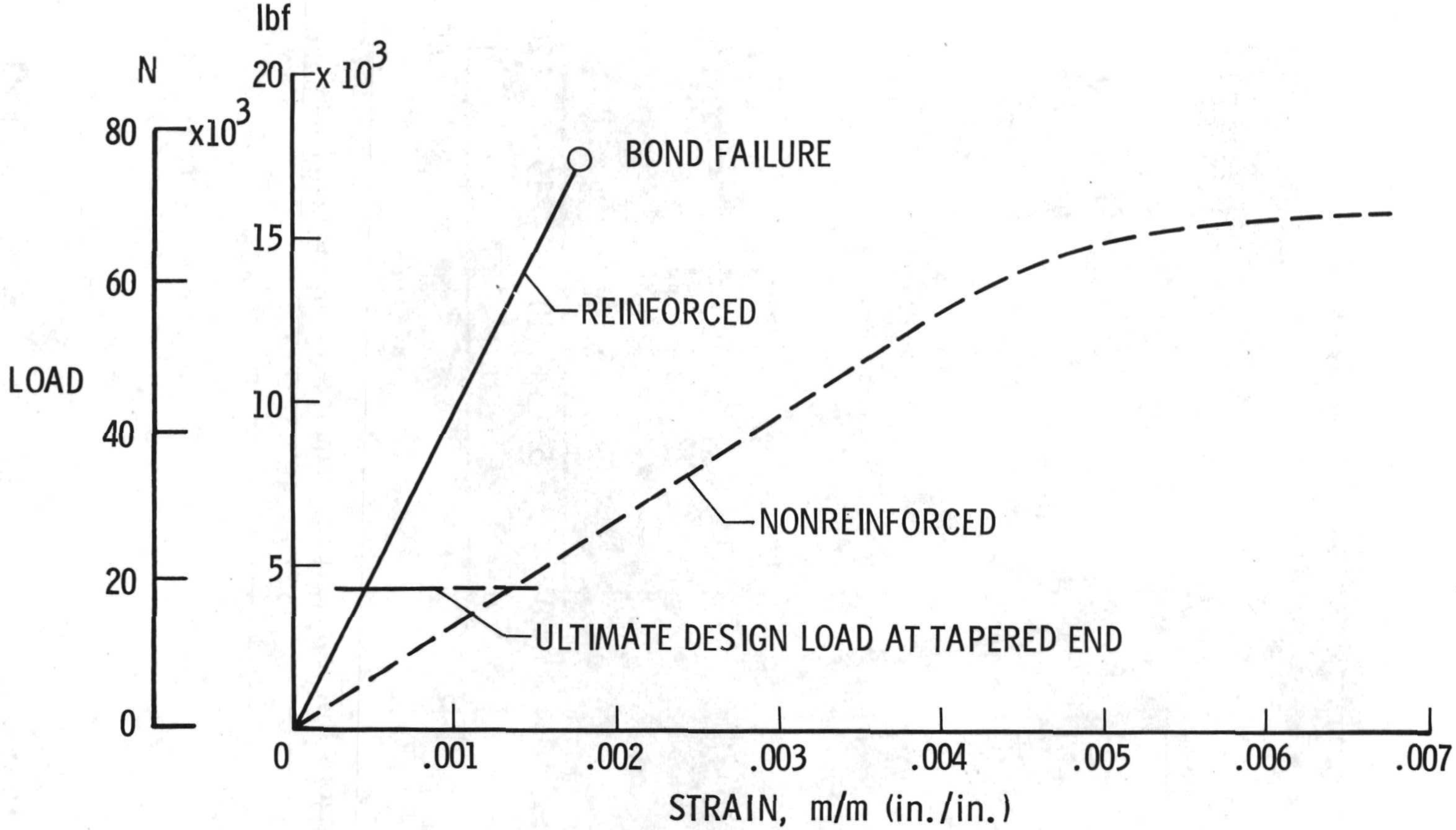
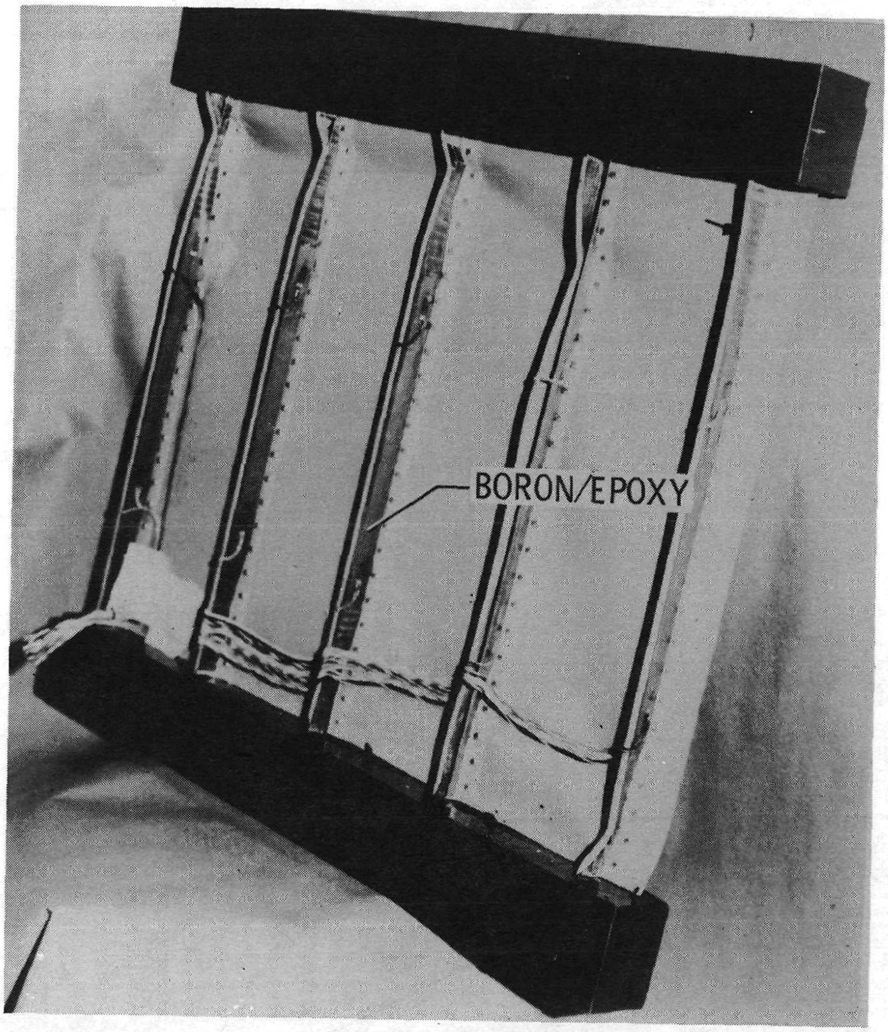


Figure 9.- Load-strain behavior for CH-54B tail-cone stringers.



L-73-3059

Figure 10.- Boron/epoxy reinforced compression panel for CH-54B tail cone.

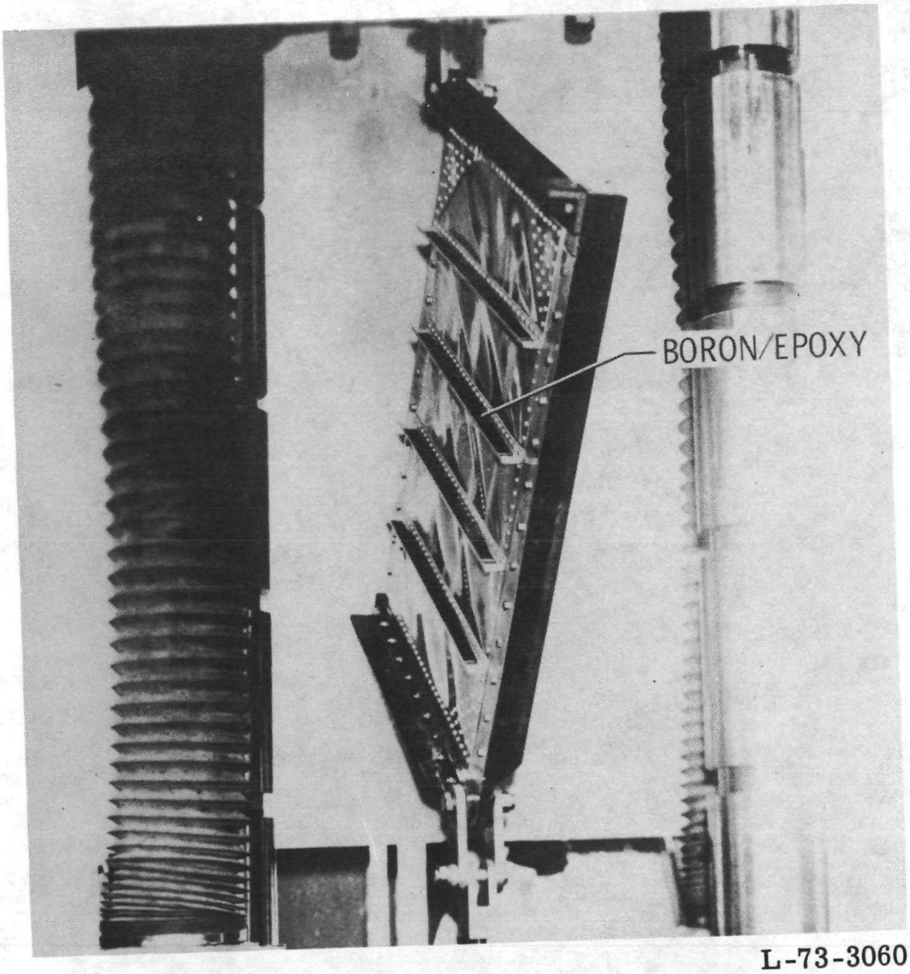


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Figure 11.- Boron/epoxy reinforced shear panel for CH-54B tail cone.

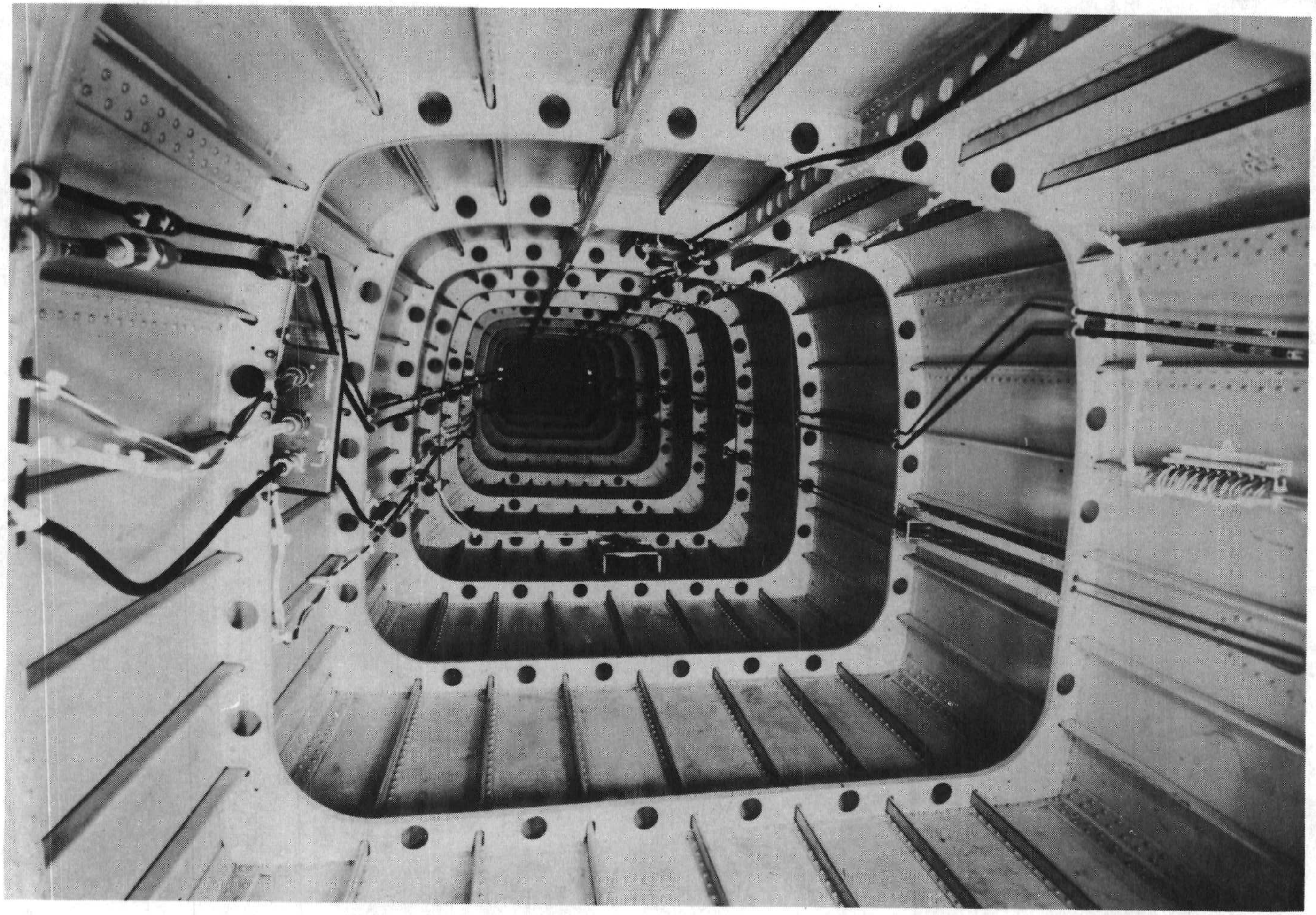

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Figure 12.- Interior of boron/epoxy reinforced CH-54B tail cone.

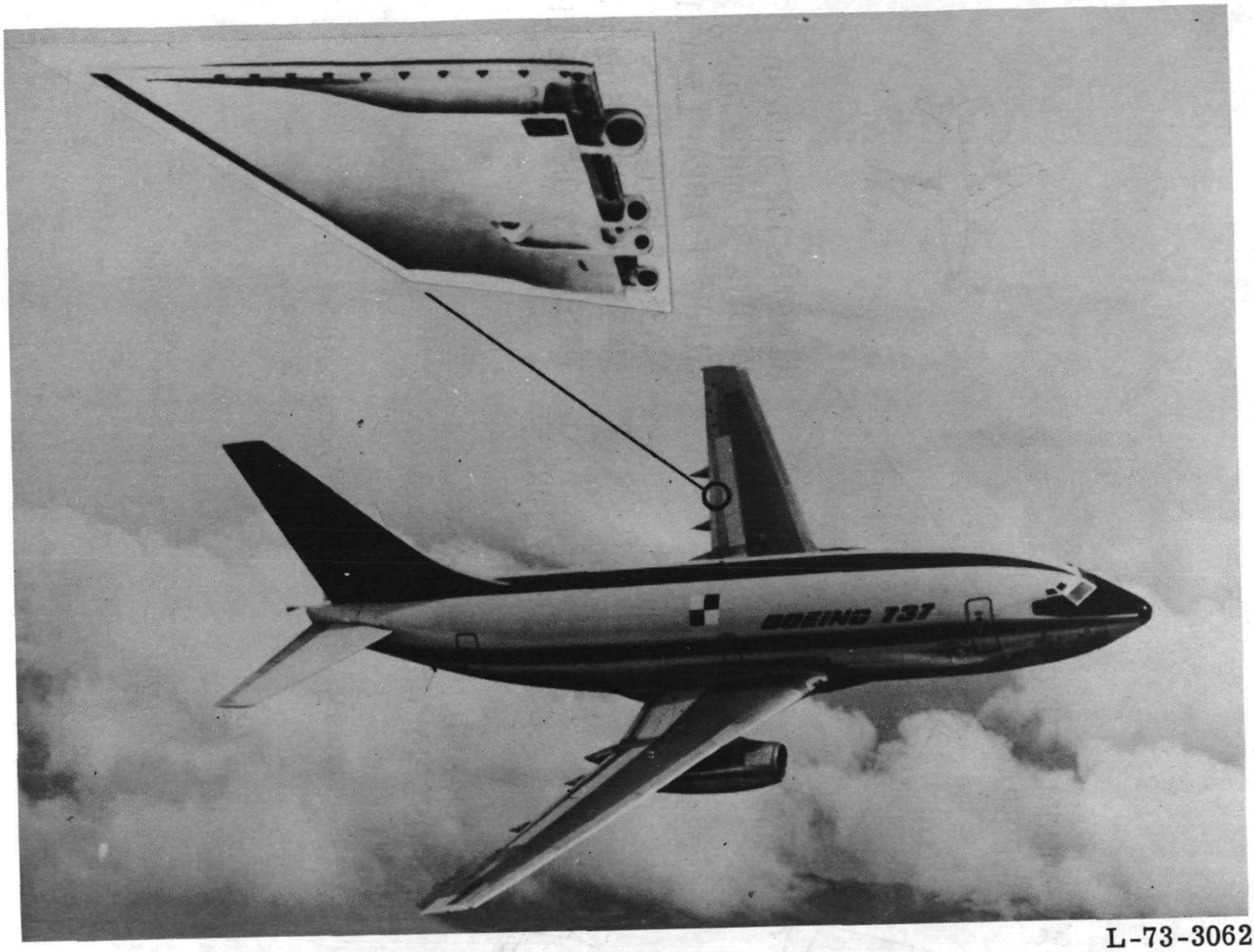

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Figure 13.- Boeing 737 composite spoilers.

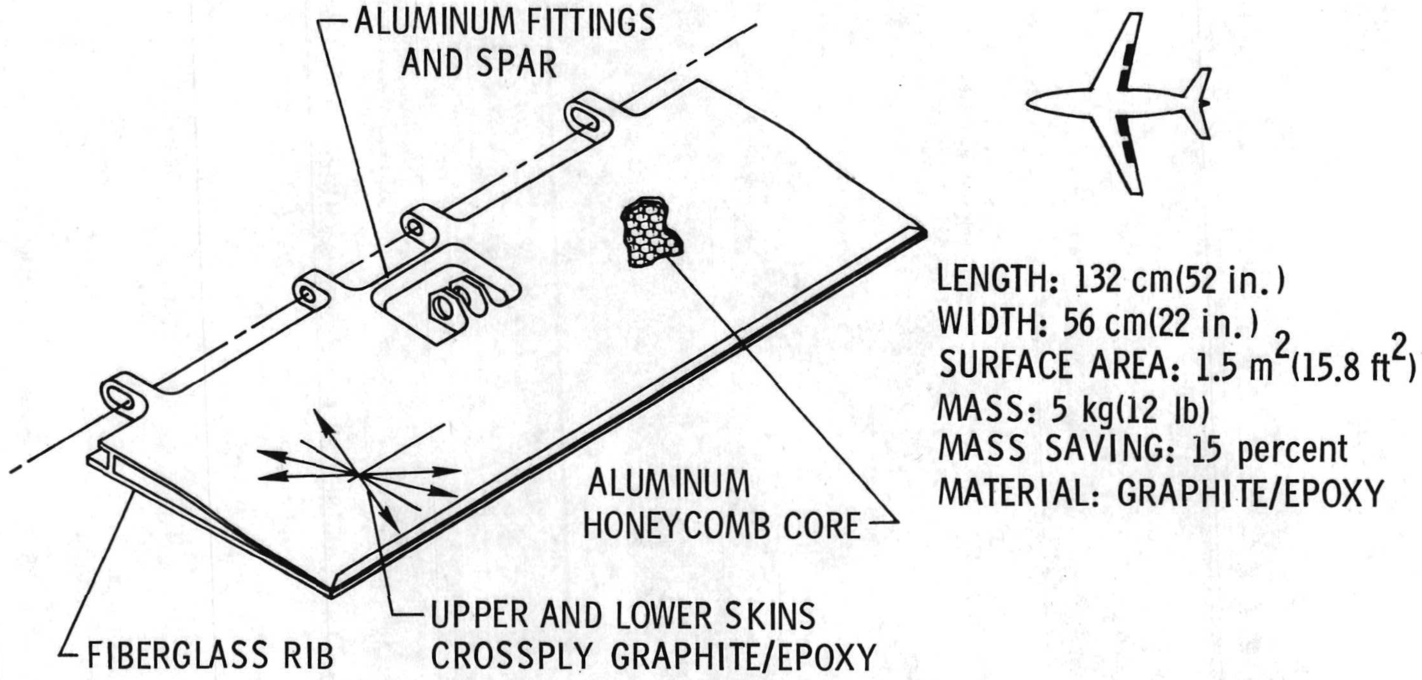

Figure 14.- Construction of Boeing 737 composite spoiler.

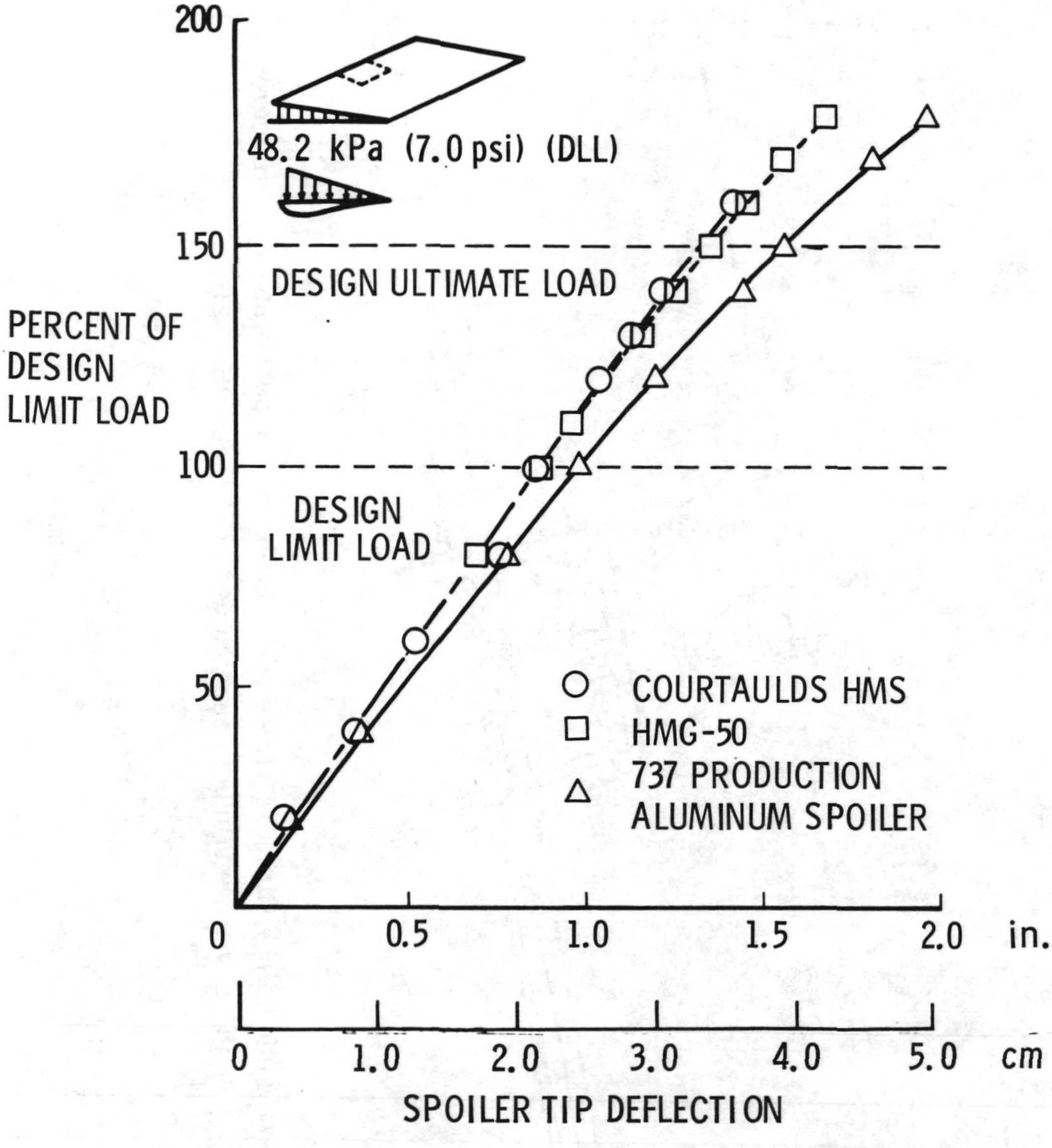

Figure 15.- Load deflection for Boeing 737 spoilers.

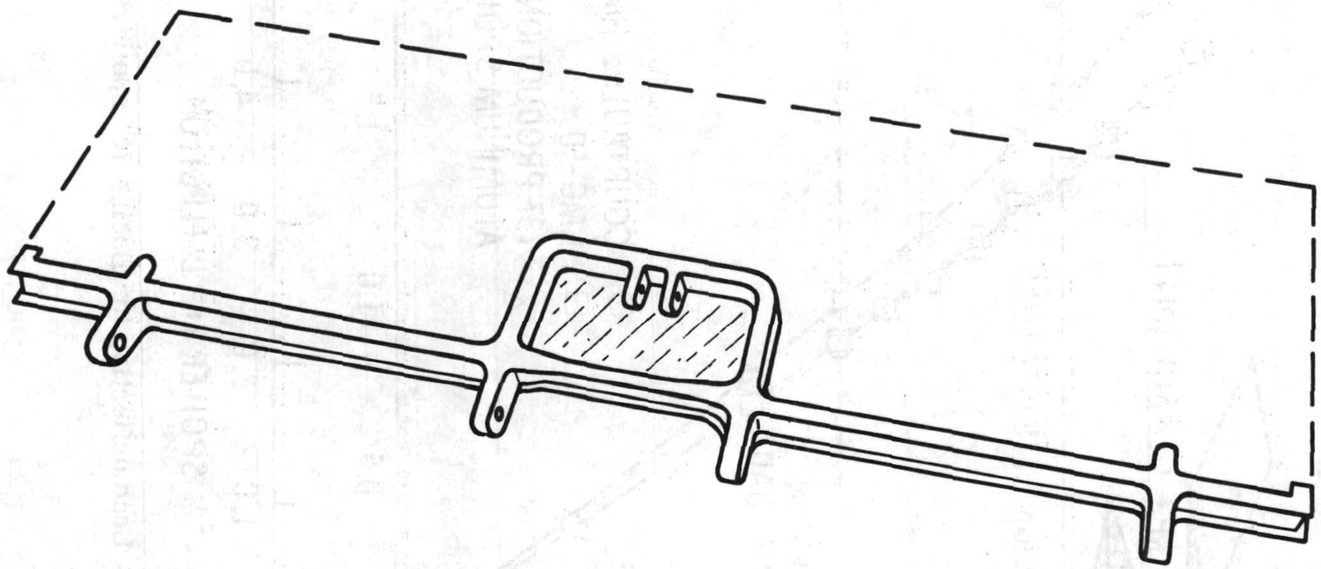

Figure 16.- Molded hinge fitting for Boeing 737 advanced composite spoiler. (Integrated fitting will contain both random and unidirectional fibers.)

L-73-3063

Figure 17.- Lockheed L-1011 PRD-49/epoxy composite fairings.

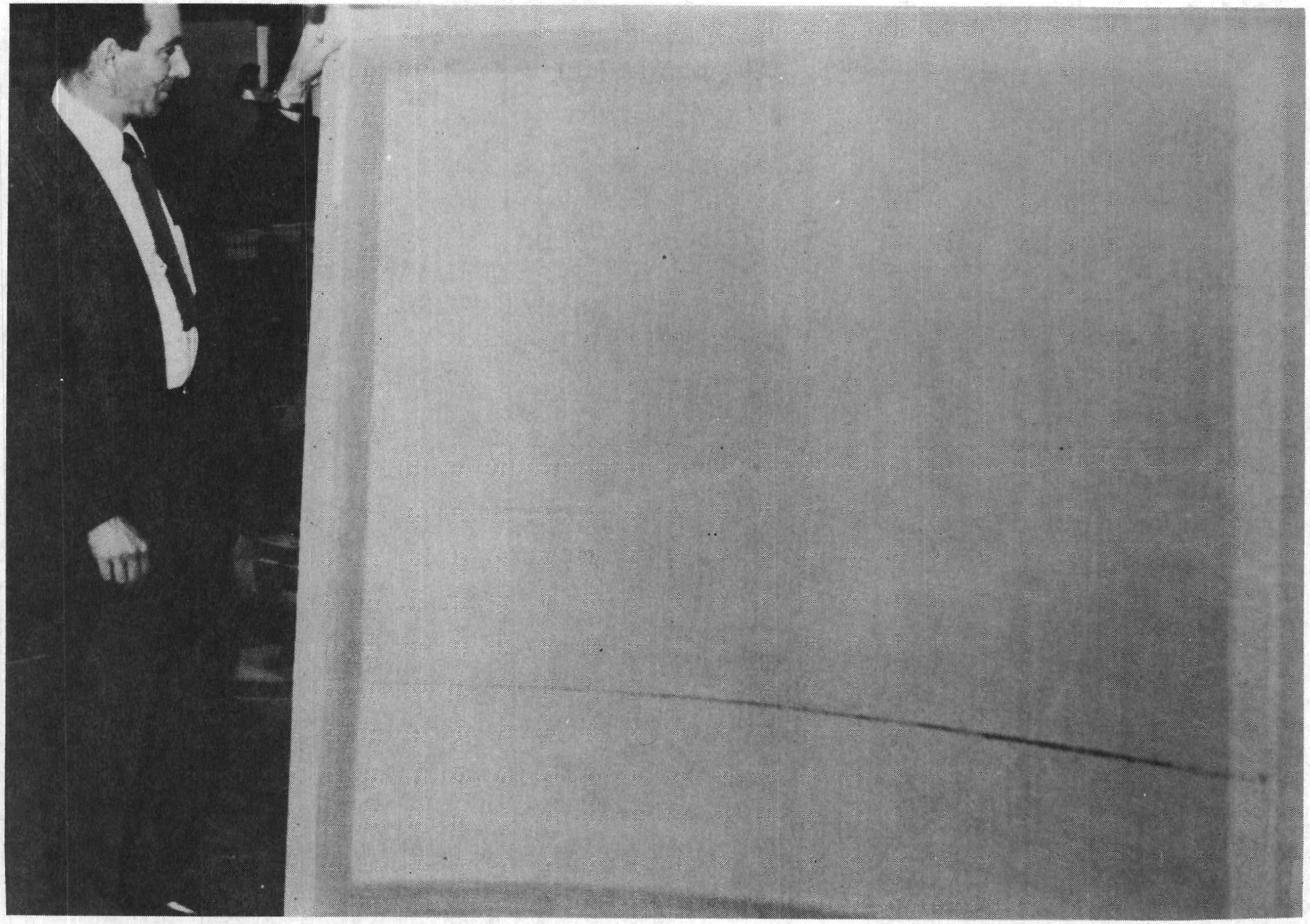

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Figure 18.- Lockheed L-1011 wing-to-fuselage fairing.

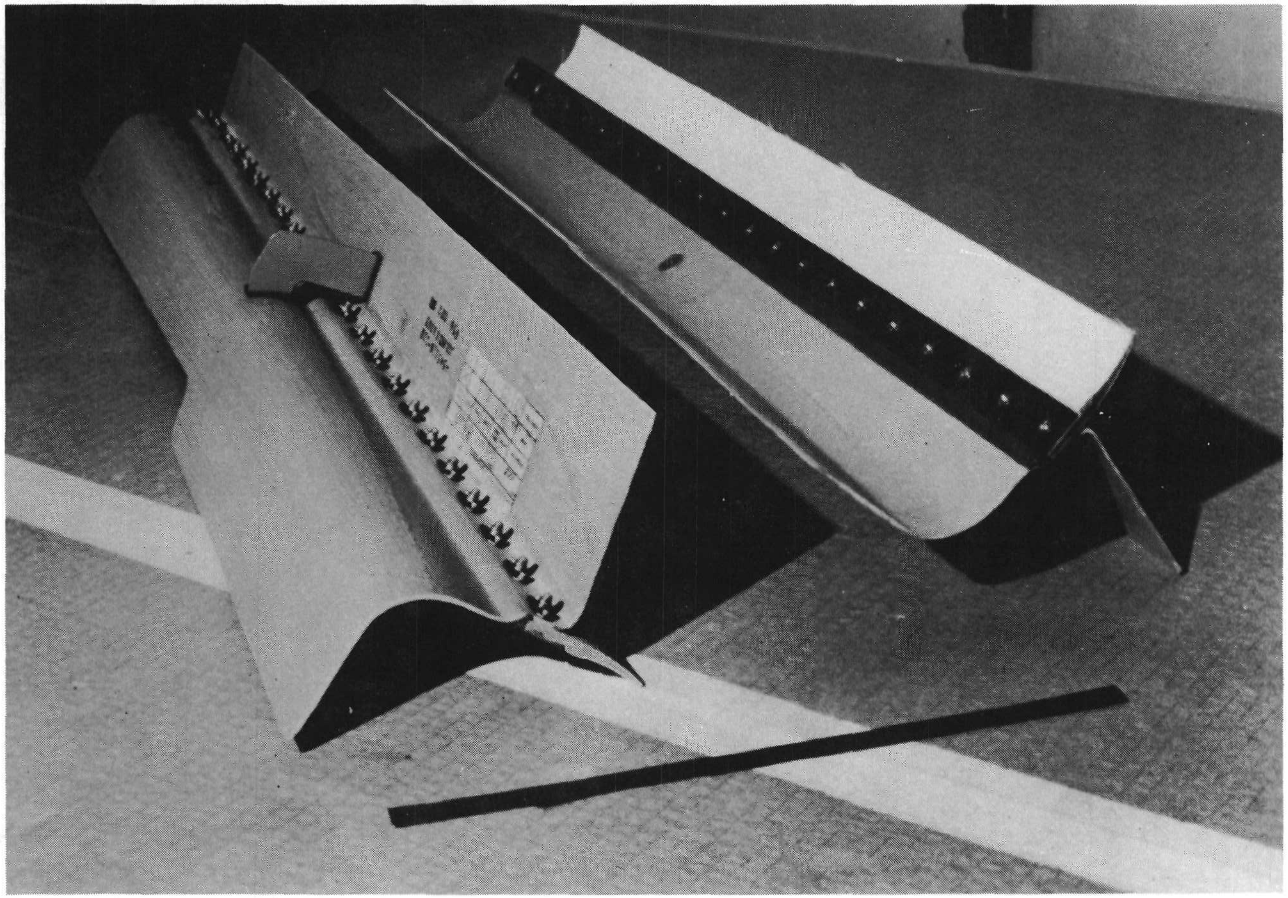

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Figure 19.- Lockheed L-1011 wing-to-fuselage fillet.

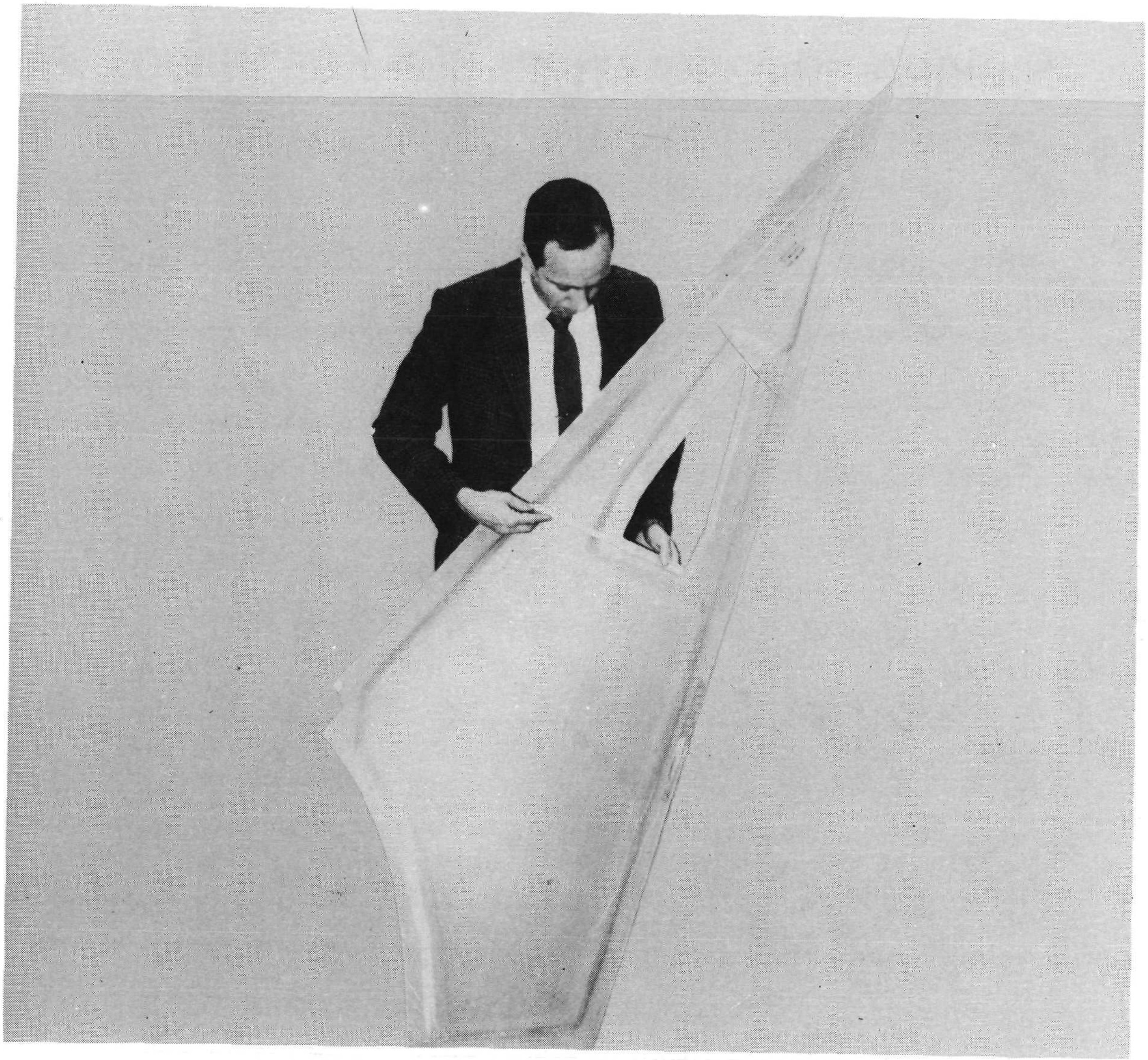

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Figure 20.- Lockheed L-1011 center-engine fairing.

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